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ADP010499 thru ADP010530

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AN OPTIMISATION PROCEDURE FOR THE CONCEPTUAL ANALYSIS OF DIFFERENT AERODYNAMIC CONFIGURATIONS

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ABSTRACT

This paper addresses the problem to define a methodology for the analysis of the performances of different aircraft configurations in the phase of conceptual design. The proposed approach is based on a numerical optimisation procedure where a scalar objective function, the take-off weight, is minimised. The optimisation algorithm has obviously important consequences both from the point of view of the computational times and of the obtained results. For this reason a preliminary discussion is made where various different methodologies are critically compared. Although the best compromise between different approaches is probably given by an integration between a genetic algorithm approach and a classical gradient method, in this phase only the latter procedure has been used to perform the simulations. The methodology takes into account the high number of geometrical parameters and the flight mechanics requirements involved in the problem.

A basic example is described, and the use of the proposed methodology to investigate the effects of different geometrical and technological parameters is discussed.

NOMENCLATURE

A	Sectional area of panel structure, m^2
AR	aspect ratio
b	span, m
c	chord length, m
c_D	global drag coefficient
c_l	sectional lift coefficient
c_L	global lift coefficient
c_{ma}	mean aerodynamic chord of the wing, m
E	efficiency (lift-to-drag ratio)
g	acceleration of gravity, m/s^2
H	cruise altitude, m
h-h_n	distance (non-dimentionalised with c_{ma}) between aircraft c.g. and neutral point
i	stabiliser angle, deg
L	length, m
$\bar{l}_t$	distance between aerodynamic centres of wing and stabiliser, m
M	Mach number
q	dynamic pressure, N/m^2
R	range at cruise velocity, km
S	surface area, m^2
S_{fc}	specific fuel consumption, (N/hr)/N
T	vertical gap, m
t/c	thickness ratio
V	cruise velocity, m/s
W	weight, N
WS	wing loading, N/m^2
x_{cg}	c.g. position of the residual weight, (% L_{fus})
x_s	position of stabiliser root leading edge from fuselage nose, m
y	spanwise coordinate, m
α	angle of attack, deg
Γ	dihedral angle, deg
δ	elevator angle, deg
θ	twist angle, positive increases the tip angle of attack, deg
λ	taper ratio
Λ	sweep angle, deg
ρ	air density, kg/m^3
ρ_M	density of structural material, kg/m^3
σ_a	admissible normal stress level, N/m^2
Φ	diameter, m

Subscripts

BL	bending due to lift
c	cruise
ea	elastic axis
f	fuel
fus	fuselage
M	pitching moment
o	take-off
p	payload
r	root
ref	reference
res	residual
rib	wing ribs
s	stabiliser surface
SL	shear due to lift
t	tip
w	wing

Paper presented at the RTO AVT Symposium on "Aerodynamic Design and Optimisation of Flight Vehicles in a Concurrent Multi-Disciplinary Environment", held in Ottawa, Canada, 18-21 October 1999, and published in RTO MP-35.

1. INTRODUCTION

During the phase of conceptual design, the problem of evaluating the performance of different aircraft configurations is of great interest; however, difficulties arise due to the high number of geometrical parameters involved, which are necessary for defining such a configuration. A systematic analysis taking into account the effects of all these parameters appears to be difficult, given the complexity related to both aerodynamic load evaluation and the assessment of flight mechanics requirements. Furthermore, a sufficiently precise estimation of the weights of the different aircraft components is necessary. However this estimation is strongly related to aircraft lift distribution, hence the use of a satisfactory aerodynamic code is mandatory.

In order to tackle this problem, a direct numerical optimisation technique may be employed, giving rise to an attractive approach, since the problem may be addressed systematically, and the designer has a great flexibility in the choice of the design variables.

In the analysis through direct numerical optimisation, an aerodynamic code is coupled in a loop with an optimisation routine, so as to automatically manage the values of the design variables – typically concerning geometry modifications – with the aim of minimising a given scalar quantity (objective function). This approach is extremely flexible, and capable of meeting multidisciplinary requirements. The choice of the design variables is of fundamental importance, as they define the set of solutions within which the optimal one is sought. Finally, the design problem formulation requires the definition of the constraint functions, which can be of aerodynamic, flight mechanics or geometrical nature, or more generally involve any quantity that may be computed with sufficient reliability.

In a previous paper[1] an aerodynamic optimisation procedure, based on a potential flow model, has been developed for the study of the effects of the geometric parameters on the induced drag. The obtained code proved to be quite reliable in terms of efficiency in determining the optimum configuration for the induced drag, but, in the version presented in Ref. 1, only this aerodynamic aspect was taken into account. Consequently, the analysed configurations did not meet the constraints resulting from aircraft flight mechanics, nor was the structural weight considered. In Ref. 2 the above obstacles were overcame by including the constraints of a realistic configuration in the code, with the assessment of trim conditions, static stability requirements and with the addition of the evaluation of airplane structural weight, based on a methodology originally developed by Torenbeek.[3] The weight was included with a low level routine, the aim being to develop a code suitable for a conceptual design approach. This simplification, however, should not be considered a real limitation in so far as different modules, which make up the code, can be substituted by other ones, with more sophisticated mathematical models, without the need to modify the problem approach.

This paper is concerned with some improvements with respect to Ref. 2. In particular, in the present version the aerodynamic code takes into account the wing thickness and the real fuselage shape. From the flight mechanics point of view, the pitching moment at zero lift has now been considered.

The whole methodology is strongly dependent of the optimisation algorithm both from the computational performance point of view and the accuracy of the obtained results. Therefore, a critical analysis of the available methods has been carried out: this is described in the next section.

2. A CRITICAL REVIEW OF OPTIMISATION METHODS

The main limitation to aerodynamic shape optimisation in a difficult context is certainly related to his prohibitive computational cost due to complex and/or three dimensional geometry, and to multi-objective problems (multipoints and/or multidisciplinary optimisation). Among the different methodologies that are being pursued for direct optimisation, gradient-based methods, in which a specified objective is minimised, are often employed. These procedures require computations of both objective functional and its derivatives respect to each control variable. The gradient calculations are usually performed using black box finite difference methods which are simply to implement but require, for each control variable, at least a cost evaluation. For practical problems, a large number of control variables are needed to take care of geometry and multidisciplinary constraints. Accordingly, when the functional is expensive to compute, this kind of method gives poor computational performances.

A first possible solution is to use relative simple aerodynamic models, as for example, potential flows (see e.g. Refs. 4-6). The adjoint approach, based on control theory or variational analysis, is an interesting alternative procedure for the calculation of the functional derivatives, which makes use of a continuous as well as a discretised differentiation (see Newman et al.[7] for a review). Actually, it allows one to greatly limit the number of cost evaluations, and thus, can significantly reduce the computational cost. For the above reasons, in the last years, several optimisation computations for complex configurations and complex flow models have been performed using the adjoint method. Usually, this technique is combined with multigrid-like algorithms in order to reduce the dependency between cost and number of control variables and/or with the powerful of parallel computing. In this way, optimisations of realistic, complete aircraft geometries, using the 3D Euler equations as flow model, have been performed[8-11] as well as 2D airfoil shape optimisations using the Reynolds averaged Navier-Stokes equations coupled with turbulence models as flow model[11-13].

The hand-code exact derivation is extremely difficult to perform due to the complexity of both equations and discretisation, thus automatic differentiation techniques[14] have been used for the evaluation of the derivatives in a discrete adjoint method[13,15]. It should be noted that methods of second-order, .i.e. methods in which not only a gradient, but also, a Hessian must be evaluated, can be found in literature (see e.g.

Ref. 16). In a context of multidisciplinary design problems, gradient methods minimise a functional constituted by a combination of different criteria with different weights. Alternative methods, which do not require the use of gradients, can be also considered, in particular stochastic ones as the genetic algorithms (GAs) (see Hajela[17] for a review). GAs, which have found significant interest for applications in the aeronautical field (see e.g. Mosetti and Poloni[18], Crosslet and Laananen[19]). They are particularly attractive in a complex context because of their robustness. Indeed, they can overcome the limitations of gradient-based methods, i.e. they can operate with very poor regularity, especially with irregular non-differentiable functions and disjointed feasible domains. They can deal with a mix of different objects as continuous, discrete, integer or boolean variables, and, they are available to approach a global optimum whereas the gradient-based methods can yield only local optima. However, the major drawback of GAs is that they require a high number of evaluations of the objective function, and thus, these methods are poorly efficient in a context of expensive cost function. In particular, once the nearness of the global optimum is reached, the non gradient methods converge very poorly whereas this condition is well-suited for gradient-based methods.

To partially overcome the lack of computational efficiency, parallel computer architectures can be used (see e.g. Refs. 20-22) drawing profit from the structure of GAs which is well adapted to a fully parallelisation of the algorithm. Gradient-based methods and GAs have complementary properties, thus, in order to combine their favourable features, an interesting approach for complex configurations as multidisciplinary aeronautics optimisation is to couple the two methods. Starting from a large area of configurations, a GA can be used to get close to a particular configuration which is, then, optimised by a gradient-based method to reach the global optimum. This kind of hybrid strategies begins to be studied as it was reported in very recent works.[23,24]

In our opinion this latter approach appears to be the most effective for the problem at hand, but in the first phase of our research only a gradient method has been applied. Actually, this method has been used to obtain the simulation results for the present paper.

3. THE MATHEMATICAL MODEL

3.1 Evaluation of aerodynamic characteristics

The pressure distribution acting on the airplane was evaluated by means of a non linear panel method, based on the formulation due to Morino; the features of this method are discussed in detail in Ref. 25.

The code allows for wake relaxation, thus yielding non-loaded stream surfaces as required by the correct boundary condition. Once the pressure distribution is known, the lift distribution over wing, fuselage and stabiliser surfaces, as well as the induced drag, can be calculated.

In consideration of the aerodynamic code chosen, the analysis is naturally confined to flows at low angles of attack and subsonic Mach numbers; nevertheless, it can provide useful information without the need for complex and expensive experimental and/or numerical investigations.

Aircraft drag, at each step of the optimisation procedure, was estimated by using the component building method. The induced drag of the configuration was predicted by means of the previously described code. The profile drag of the lifting surfaces was evaluated by means of standard preliminary methodologies, as described in Refs. 26 and 27, while the drag of the non lifting bodies (fuselage and vertical tail) was evaluated following similar methodologies.[26,28] The drag increment caused by the propulsive system installation is very sensitive to the choice of a specific configuration; since in this phase no consideration at all is made on propulsion system choice and position, it was preferred to avoid the estimation of its effects. Consequently, the total drag is slightly underestimated, even if this simplification does not significantly affect the optimisation procedure. Obviously, there are no obstacles in taking into account this quantity where more detailed studies are performed.

3.2 The numerical optimisation routine

In the analysis through direct numerical optimisation,[29] an aerodynamic code is coupled in a loop with an optimisation routine, so as to automatically manage the values of the design variables – typically concerning geometry modifications – with the aim of minimising a given scalar quantity (objective function). This approach is extremely flexible, and capable of meeting multidisciplinary requirements. The choice of the design variables is of fundamental importance, as they define the set of solutions within which the optimal one is sought. Finally, the design problem formulation requires the definition of the constraint functions, which can be of aerodynamic, flight mechanics or geometrical nature, or more generally involve any quantity that may be computed with sufficient reliability.

It should be emphasised that a numerical procedure of this kind can never be regarded as a completely automated tool; user's experience and external control on the process are always needed to obtain the best results. In fact, the topology of the feasible region is unknown a priori.

The numerical optimisation routine adopted for the present study is the CONMIN code,[30] used extensively in the fields of aerodynamic and structural optimisation. CONMIN uses a gradient method for the search algorithm and includes three alternative first order methods for the calculation of the vector search direction. When all the constraints are satisfied for the current values of the design variables, the steepest descent method is used for the first iteration, and the conjugate direction method for subsequent iterations. When some of the constraints are active or violated the feasible directions method is used.

3.3 The optimisation procedure

In this paper the main objective is to evaluate the capabilities of the proposed methodology to obtain configurations with improved performances with respect to a reference one. The analysis is carried out upon assuming that the following design parameters have been arranged for cruise: payload, velocity, range, cruise height, engine specific fuel consumption. We stress that the different configurations were designed only for achieving a high-speed, long range cruise mission by matching the airplane design to cruise performance design. No dynamic analysis nor high lift conditions were directly considered. However, the high lift condition is implicitly inserted in the design trough a constraint on the wing loading, as discussed in the following.

It has been supposed that payload establishes the length and diameter of the fuselage, which have been considered known. Thickness ratio, sweep and dihedral angles for both wing and stabiliser, and the ratio between the wing and tail surfaces have been assumed as design parameters and have been kept fixed. As to tail volume (imposed here by means of the ratio between wing and tail surfaces), this is obviously imposed by mission segments different from cruise (typically, take-off and landing). The sweep angles are imposed by the Mach number, while dihedral angles comes from lateral stability considerations. Finally, as to thickness ratio, this is a function of both cruise and high-lift conditions. Cruise imposes a maximum *t/c* owing to drag problems; as to high lift conditions, it can be easily shown that the function c_{Lmax} versus *t/c* has a maximum which depends on the wing section. This entails another constraint on the maximum value of *t/c*. The choice of some of the above design parameters may have significant effects on the results; accordingly, the influence of a variation in thickness ratio and tail volume coefficient could be investigated, as shown in Ref. 2.

For each lifting surface, the design variables used in the optimisation process are aspect ratio, taper ratio and twist. Moreover, wing loading and stabiliser position (horizontal stagger and vertical gap) are also design variables. However, wing loading is a variable which is typically dependent upon flight phases other than cruise and, therefore, it is not possible to optimise this variable without a constraint which forbids it to assume unrealistic values. Moreover, stabiliser position is constrained by fuselage geometry, since its length and diameter have been fixed.

It is well known that one of the crucial aspects of the optimisation procedure is constituted by the choice of the object function. In the present analysis various considerations lead us to the conclusion that the take-off weight is a reasonable choice, even if it should be clear that the essence of the proposed methodology would not be affected by a different objective function. The main reason is that the aircraft take-off weight appears to offer a quite reasonable compromise between the need to obtain solutions that are sufficiently accurate to give significant results and the need to prevent such solutions from being excessively complicated, as it would happen by using, for example, Direct Operating Costs as an objective function.[31]

The objective function is considered to be the sum of four terms: wing weight, stabiliser weight, fuel weight and residual weight. The latter comprises fuselage, payload, vertical tail, engines, gears and various systems; their weights and centre of gravity positions are assumed to be known, and have been established by statistical data.

The optimisation process includes the following steps:

1. On the basis of the above design parameters and with a fixed set of design variables, a first estimation of W_o and W_f is given and, as a function of W_o and W_f, an estimation of W_w and W_s is obtained by using formulae derived from statistical data.[32] Accordingly, wing and stabiliser planform is fixed as a function of the distance between wing and stabiliser aerodynamic centres. Centre of gravity of wing, stabiliser and fuel (supposed to be concentrated in the wing) are also evaluated. In the following examples the positions of the centres of gravity of various elements are obtained by means of simplified formulae.[33]

2. The angle of attack and elevator angle are computed to trim the aircraft in straight, horizontal flight, by using the following equations

   $$c_{L_{trim}} = \frac{2W_o}{\rho_c S_w V_c^2} = c_L\big|_{\alpha=0,\,\delta=0} + c_{L_\alpha}\alpha + c_{L_\delta}\delta \qquad (1)$$

   $$c_M = 0 = c_M\big|_{\alpha=0,\,\delta=0} + c_{M_\alpha}\alpha + c_{M_\delta}\delta \qquad (2)$$

   The aerodynamic coefficients in Eqs. (1) and (2) are computed with an iterative procedure, by using the previously described non-linear aerodynamic code. The value of c_{Dtrim} is then evaluated with the previously described aerodynamics procedure and vehicle efficiency is obtained by $E=c_{Ltrim}/c_{Dtrim}$.

3. By means of the computed values of $c_{L\alpha}$ and $c_{M\alpha}$ the static longitudinal stability of the configuration is checked. The code constrains the difference between centre of gravity and aerodynamic centre positions (non-dimentionalised by c_{ma}) to be included between two imposed bounds. In the examples the inequality $-0.05 \le (h-h_n) \le -0.2$ was used. When the above constraint is violated, the code repetitively comes into play to modifying the value of l_t, by moving the wing with respect to fuselage, until the bounds are met.

4. At this point, the load distribution on lifting surfaces is known, therefore it is possible to give a new estimation of wing weight by using the methodology of Torenbeek[3] which binds wing weight to lift distribution along the wing span and mechanics characteristics of the employed materials (see Appendix). By this means it is possible to accurately estimate wing weight.
   Fuel weight is obtained from Breguet's formula:

$$W_f = W_o\left[1-\exp\left(-\frac{RS_{fc}}{V_c E}\right)\right] \tag{3}$$

and the quantity $\overline{W} = W_w + W_s + W_f + W_{res}$ is evaluated. If $\overline{W} \neq W_o$ (except for numerically insignificant differences), a new estimation of W_o is given by using, for example, the updating formula $W_o' = (\overline{W} + W_o)/2$.

5. The points 1-4 are repeated until the condition $\overline{W} = W_o$ is met. It should be noted, however, that W_w at point 1 is now evaluated by Torenbeek's approach.

6. The optimisation routine updates the design variables and the procedure described at points 1-5 is repeated until the minimum weight configuration is obtained.

We stress the fact that, since the wing weight is evaluated by means an approach based on the real spanwise lift distribution, (and not simply on statistical data) it is possible to analyse canard configuration as well. Nevertheless, in this case the estimations based upon a statistical approach are defective due to the small number of available data.

4. THE OPTIMISATION PROCEDURE: A CASE STUDY

In this section the outlined methodology has been applied, as an example, to a typical light transport aircraft. Table 1 shows the values of the fixed parameters used in the optimisation procedure.

W_p	x_{cg}	M	R	S_{fc}	H
33000	40	0.4	2100	0.48	6100

L_{fus}	Φ_{fus}	Λ_w	Λ_s	Γ_w	Γ_s
11.2	1.7	0°	4°	4°	9°

Table 1 - Fixed parameters used in the optimisation procedure

In addition to the data of Table 1, the following design parameters have been fixed: (t/c) from 0.18 (at the root) to 0.12 (at the tip), both for wing and tail. Moreover a ratio S_t/S_w=0.25 has been fixed.

The design variables are aspect ratio, taper ratio and twist of wing and tail, wing loading and tail position. The constraints are as follows: 1) maximum value of wing loading WS=1903 N/m^2; 2) the tail vertical position could only vary between the maximum height of the fuselage and two meters above that position; 3) tail horizontal position constrained by fuselage length. Note that a fully rotating tail has been assumed. It is interesting to note that the objective function was highly sensitive to wing loading which reached the value given by the constraint. On the other hand, the optimisation of tail variables turned out to have a negligible effect, and the tail position reached, in all the cases presented, the upper values of the constraints. The most important results of the optimisation process have been summarised in Table 2a, where aircraft geometry is defined, and Table 2b, where trim conditions are shown.

	Wing	Tail
S	23.46	5.87
b	15.14	5.97
c_r	1.80	1.31
AR	9.76	6.07
λ	0.72	0.5
θ	-2.64°	0°
l_t	5.46	
x_S	10.9	
T	2.00	
i	0°	
W_w	5461	
W_s	514	
W_f	5686	
W_o	44652	

Table 2a - Aircraft geometry and weights

α	1.52°
δ	1.37°
c_L	0.361
c_D	0.0218
E	16.6

Table 2b - Trim conditions

The geometry of the optimised configuration is shown in Fig. 1 along with the mesh used in the computational analysis (for the body representation, about 2700 panels have been used). Typical computational times required to obtain an optimised configuration are of the order of 5h on a Pentium III Xion, 500 MHz and 512 MB RAM.

As far as drag analysis is concerned, wing loading being practically fixed by the constraint, it can be seen that wetted area is practically a constant (it is clear that, for a correct optimisation process, the final take-off weight cannot be significantly different from the initial estimation), hence friction drag does not vary to any remarkable extent and the optimisation process tends almost exclusively to decrease the induced drag. Accordingly, the optimised configuration is the best compromise between two opposite standpoints: an increase in wing span to decrease induced drag (thereby decreasing W_f and W_o) or, conversely, a reduction in wing span to reduce structural stress and, as a result, to diminish W_w and W_o.

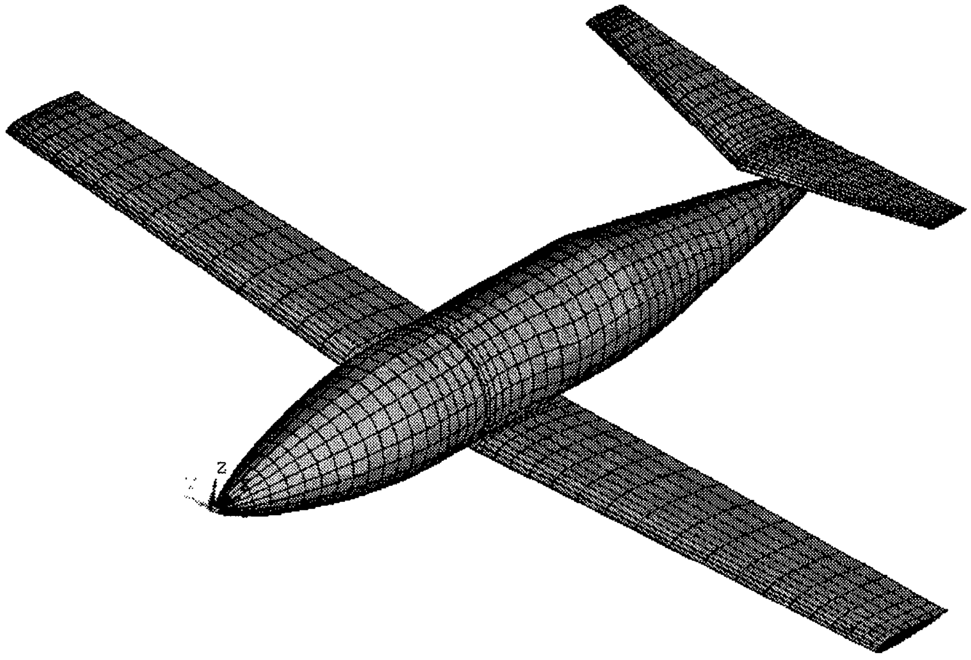

Fig. 1 – Reference optimised configuration

5. A COMPARISON OF DIFFERENT CHOICE OF THE GEOMETRICAL PARAMETERS

Although the above parameters are constrained, from a practical point of view, by mission segments different from cruise (especially take off and landing), it is interesting to obtain quick estimates of the order of magnitude of the improvements which would be related to a relaxation of these constraints. This in turn gives important information about the trade-off between an increase in costs related to more sophisticated solutions and an increase in the overall configuration performance.

The optimised configuration outlined in the previous section results from a number of different geometrical constraints. It is interesting to investigate the effect of the parameters which are not directly controlled by the optimiser. In particular, we analyse the differences in the configuration due to a different choice of the wing thickness ratio and to a different wing loading.

In Table 3 the optimised configuration obtained with a greater wing loading constraint is compared to the reference one.

A significant reduction in wing weight with an increase in wing loading is apparent, along with a less important effect on fuel consumption. Note that the increase in wing loading causes a decrease in wing span and an slight increase in aerodynamic efficiency, thus leading to both a reduction in wing weight and lower fuel consumption.

Value	WS=1903(Ref)	WS=2093
S_w	23.46	20.93
b_w	15.14	14.04
AR_w	9.76	9.41
λ_w	0.72	0.72
θ_w	-2.64	-4.64
b_s	5.97	5.64
λ_s	0.5	0.5
θ_s	0	0
l_t	5.46	5.14
c_L	0.361	0.396
c_D	0.0218	0.0237
E	16.6	16.7
W_w	5461	4752
W_s	514	461
W_f	5686	5549
W_o	44652	43813
$(W_o - W_{oref}) / W_{oref}$	-	-1.88%

Table 3 – Effect of a different wing loading

In Table 4 the optimised configuration obtained with a reduced value of thickness ratio at wing root is compared to the reference one. In particular, the reference value, corresponding to t/c=0.18 at wing root, is modified in t/c=0.16.

Value	t/c_{root}=0.18 (Ref.)	t/c_{root}=0.16
S_w	23.46	23.54
b_w	15.14	15.27
AR_w	9.76	9.90
λ_w	0.72	0.71
θ_w	-2.64	-2.68
b_s	5.97	5.98
λ_s	0.5	0.5
θ_s	0	0
l_t	5.46	5.34
c_L	0.361	0.360
c_D	0.0218	0.0220
E	16.6	16.4
W_w	5461	5505
W_s	514	524
W_f	5686	5763
W_o	44652	44806
$(W_o - W_{oref})/ W_{oref}$	-	+0.34%

Table 4 – Effect of a different thickness ratio

It is clear that weight increases as *t/c* at the root decreases. Upon decreasing *t/c* at the root, performance decreases due to an increase in wing weight. It should be noted, however, that the higher structural efficiency is not "used" to increase wing span, which, indeed, shows a slight decrease. When *t/c* at the root is increased from 0.16 up to 0.18, the higher structural efficiency is "used" to increase wing span, thus reducing the induced drag, with a consequently lower fuel consumption, but without important differences in wing weight.

6. A COMPARISON OF DIFFERENT CHOICE OF THE TECHNOLOGICAL PARAMETERS

In this section, the outlined methodology has been applied to stress the importance of some of the most significant technological parameters over the final aircraft configuration.

6.1 Reduction in friction drag

It is known that the reduction of friction drag plays a relevant role in the current aerodynamic research. In what follows, we suppose that the friction drag may be reduced by 10% with respect to the nominal case. As a first example, this reduction has been applied to the optimised configuration of Table 2, which results in a fuel reduction of 370 N, which in turn corresponds to a reduction of 0.8% with respect to the take off weight.

It is interesting to investigate if the optimisation process would be different in this case: accordingly the whole optimisation procedure has been carried out with the new value of the friction coefficient and the results are shown in Table 5.

The optimised configuration obtained considering a smaller friction drag shows a reduction in the take off weight of 1.33%, significantly higher (+0.53%) than the corresponding result obtained considering the reduction in friction drag applied to the reference configuration.

Value	Reference	c_f=0.9$c_{f,ref}$
S_w	23.46	23.15
b_w	15.14	14.64
AR_w	9.76	9.26
λ_w	0.72	0.74
θ_w	-2.64	-4.37
b_s	5.97	5.93
λ_s	0.5	0.5
θ_s	0	0
l_t	5.46	5.38
c_L	0.361	0.360
c_D	0.0218	0.0206
E	16.6	17.5
W_w	5461	5166
W_s	514	505
W_f	5686	5330
W_o	44652	44057
$(W_o - W_{oref})/ W_{oref}$	-	-1.33%

Table 5 – Effect of a different friction drag

The new configuration shows an increased aerodynamic efficiency, even with a reduced wing span; therefore, both the wing and fuel weights are reduced. The lift coefficient remains practically constant, therefore the aerodynamic flow is practically the same, while the geometric configuration appears slightly different. Indeed, the new configuration has a reduced wing surface with a reduction both in wing span (-3.3%) and aspect ratio (-5.1%), a slightly higher taper ratio and an increased twist.

6.2 Reduction in specific fuel consumption

Another fundamental research area in the aeronautical field relates to the increase in engine efficiency. Accordingly, in what follows we suppose that the specific fuel consumption may be reduced by 10% with respect to the nominal case. As a first example this reduction has been applied to the optimised configuration of Table 2, which results in a fuel reduction of 569 N, which corresponds to a reduction of 1.3% with respect to the take off weight. Obviously, in this case the reduction in fuel weight is very close to the corresponding reduction in specific fuel consumption.

As in the previous example, the whole optimisation procedure has been carried out with the new value of the new value of the specific fuel consumption and the results are shown in Table 6.

The optimised configuration obtained considering a smaller specific fuel consumption shows a reduction in the take off weight of 1.98%, significantly higher (+0.68%) than the corresponding result obtained considering the reduction in specific fuel consumption applied to the reference configuration. The reduction in specific fuel consumption directly reduces the fuel weight, with a consequent reduction in the wing weight, mainly caused by the reduction in required total lift.

Value	Reference	S_{fc}=0.9S_{fcref}
S_w	23.46	23.00
b_w	15.14	14.59
AR_w	9.76	9.26
λ_w	0.72	0.74
θ_w	-2.64	-4.55
b_s	5.97	5.91
λ_s	0.5	0.5
θ_s	0	0
l_t	5.46	5.38
c_L	0.361	0.360
c_D	0.0218	0.0220
E	16.6	16.4
W_w	5461	5135
W_s	514	501
W_f	5686	5119
W_o	44652	43767
(W_o- W_{oref})/ W_{oref}	-	-1.98%

Table 6 – Effect of a different S_{fc}

The new configuration shows an aerodynamic behaviour practically unchanged with respect to the reference configuration: the lift coefficient, the drag coefficient and the efficiency remain practically constant, therefore the aerodynamic flow is almost the same, while the geometric configuration appears to be slightly different. Indeed, the new configuration has a reduced wing surface with a reduction both in wing span (-3.3%) and aspect ratio (-5.1%), a slightly higher taper ratio and an increased twist.

6.3 Increase in structural efficiency

As a final example, we investigate the effect of an increase of 10% of the admissible normal stress level of the material over the optimised configuration. Upon applying this modified value to the optimised configuration of Table 2, a reduction of wing weight of 277 N is obtained along with a reduction of 35 N of fuel weight, which corresponds to a reduction of 0.6% with respect to the take off weight.

The whole optimisation procedure has been carried out with the new value of σ_a and the results are shown in Table 7.

The optimised configuration obtained considering a higher structural efficiency shows a reduction in the take off weight of 0.93%, higher (+0.33%) than the corresponding result obtained considering the increase in structural efficiency applied to the reference configuration, but not so important as in previous analysed cases.

The new configuration shows an aerodynamic behaviour practically unchanged with respect to the reference configuration: the lift coefficient, the drag coefficient and the efficiency remain practically constant, therefore the aerodynamic flow is almost the same. The geometric configuration appears substantially unchanged, with an increased twist as the only significant modification.

Value	Reference	σ_a =1.1 $\sigma_{a\,ref}$
S_w	23.46	23.24
b_w	15.14	15.08
AR_w	9.76	9.79
λ_w	0.72	0.72
θ_w	-2.64	-4.49
b_s	5.97	5.94
λ_s	0.5	0.5
θ_s	0	0
l_t	5.46	5.29
c_L	0.361	0.360
c_D	0.0218	0.0218
E	16.6	16.5
W_w	5461	5024
W_s	514	508
W_f	5686	5656
W_o	44652	44235
(W_o- W_{oref})/ W_{oref}	-	-0.93%

Table 7 – Effect of a different σ_a

Therefore, the reduction in the take off weight is mainly produced by the direct effect of the increased structural efficiency on the wing weight and a consequent limited effect on the fuel weight.

7. CONCLUDING REMARKS

A methodology for the analysis and comparison of the performance of different configurations in the phase of aircraft conceptual design has been developed. In the analysis, a scalar objective function, the take-off weight, is minimised by means of a numerical optimisation technique, which takes into account the high number of geometrical parameters and the flight mechanics requirements involved in the problem. The approach proved to be extremely flexible, and capable of really meeting multidisciplinary requirements.

The obtained results highlight the order of magnitude of the improvements of the objective function due to an increase in technological characteristics. Moreover, it is clear that significantly higher improvements in the performances can be obtained if the configuration is developed by taking into account the above technological improvements from the initial phase of the project. This means that the optimised final configuration is different from the reference one (which in turn has been optimised with different values of these parameters).

We also note that the obtained results are "close" to the reference configuration. This is due to the fact that a gradient based optimiser may only give local minima. Actually, better results may be obtained by combining a global (GA based) optimiser, with a local gradient based approach. This matter is currently under development.

It is clear that the obtained results are not definitive, since a significant number of different configurations must be considered and several points need to be verified with a more accurate analysis. Among them the most important one is the high lift condition which was not considered directly in the optimisation process, even if we assumed that it imposes a wing loading constraint. Obviously, this constraint must be accurately verified with an a posteriori analysis.

8. APPENDIX

According to Torenbeek's approach,[3] basic wing weight is considered as the sum of three contributions: 1) the weight of material required to resist bending, W_{BL} 2) the weight of material required to resist shear forces, W_{SL} and 3) the weight of wing ribs, W_{rib}. All three contributions are calculated as follows.

8.1 Weight due to bending moment

The bending moment due to lift at a generic (y) wing station is given by:

$$M_{BL}(y) = \int_y^{b/2} \frac{y'-y}{\cos\Lambda_{ea}} q c_l(y') c(y') dy'$$

and $c_l(y')$ is evaluated by the aerodynamic code.

The basic weight of two skin panels (upper and lower) can be estimated by the formula:

$$W_{BL} = 2\rho_M g \int_0^{b/2} A(y) dy$$

with $A(y)$ given by:

$$A(y) = \frac{M_{BL}(y)}{\eta_t t(y) \sigma_a}$$

where σ_a is the given admissible normal stress level, t(y) is the maximum thickness of the profile section and η_t is an efficiency factor which may be obtained from a drawing of the wing cross section or by simplified expressions.[3]

8.2 Weight due to shear forces

The weight due to shear forces can be expressed as

$$W_{SL} = \frac{2\rho_M g}{\bar{\tau}} M_{BL}(y=0)$$

where $\bar{\tau}$ is the mean value of shear stress which has been assumed constant. $\bar{\tau}$ can be estimated by means of the simple relationship:

$$\bar{\tau} = 0.5\sigma_a$$

8.3 Weight of wing ribs

Since a rational derivation for the weight of ribs is not feasible, the following (statistical) formulation is used:

$$W_{rib} = k_r \rho_M g S \cdot [t_{ref} + (t_r + t_t)/2]$$

where k_r =0.5x10^{-3} and t_{ref} =1.0 m.

Finally, basic wing weight estimation is the sum of the above three contributions, hence:

$$W_w = W_{BL} + W_{SL} + W_{rib}$$

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